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# NASA CONTRACTOR REPORT

NASA CR-308

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# STUDY OF HEAT SHIELDING REQUIREMENTS FOR MANNED MARS LANDING AND RETURN MISSIONS



SUMMARY REPORT

Perpared under Contract No. NAS 2-1798 *by*
LOCKHEED MISSILES AND SPACE COMPANY
Sunnyvale, Calif.



*for*

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION • WASHINGTON, D. C. • OCTOBER 1965

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## 1.0 INTRODUCTION

Manned exploration missions to Mars may involve aerodynamic braking from hyperbolic approach speeds both at Mars and Earth. Atmospheric penetration at the anticipated approach velocities will subject entry vehicles to extreme thermal environments. The associated heat shielding requirements will be of major importance in selecting an optimum mission course. In the hyperbolic entry situation, complicated interactions exist among the aerothermal phenomena that are commonly neglected or at least treated independently. Significant advances in the aerothermal technology are required to ensure efficient and reliable heat-shield designs.

The problem of defining thermal protection requirements for Mars-mission entry vehicles was studied by Lockheed Missiles & Space Company for NASA under Contract NAS 2-1798. This report describes the scope of the study and briefly summarizes the principal results. A comprehensive review of the investigation is presented in the final report.

## 2. STUDY OBJECTIVES

The primary purpose of the study was the parametric determination of heat shielding requirements for manned entry at both Earth and Mars considering as variables entry velocity, vehicle volume, and vehicle weight. Secondary objectives were to delineate major technical problems and to indicate associated uncertainties in heat shield design. Subsidiary objectives were as follows:

- Study atmospheric characteristics and high temperature gas properties
- Define entry trajectories and establish aerodynamic limits on entry conditions
- Determine vehicle flow fields and heat transfer distributions
- Investigate performance characteristics of candidate heat-shield materials
- Define heat-shield designs and establish their thermal responses
- Evaluate the effects of uncertainties on shield designs

The vehicle configurations, entry situations, and parameter ranges to be considered in the investigation are summarized in Table I.

3. RELATIONSHIP TO NASA PROJECTS

The study was performed for the Ames Research Center of the NASA. It is a part of a continuing program of interplanetary mission studies. The results have direct application in the analysis of the Mars landing mission feasibility. Furthermore, they enable identification of problems for future research and they provide a basis for directing design optimization studies. The basic data generated and the analysis methods developed have general application in planetary entry work.

4. METHOD OF APPROACH

Previous investigations of the significant physical phenomena associated with hyperbolic speeds have been limited. Consequently the initial effort in the study was devoted to the formulation of analysis procedures and to the computation of basic properties data. An attempt was made to place the proper emphasis on all aspects of the problem and to employ the most rigorous techniques possible within the scope of study. The analysis procedures that were developed are sufficiently complex so as to require a high degree of automation in their application. To this end, all major computations were coded for digital computer solution and an efficient information flow process was developed. Even so, the number of parameters that must be considered in determining heat shielding requirements was large and required a careful scheduling of select cases within the wide spectrum of possibilities. Nylon phenolic, an efficient state-of-the-art ablative material, was adopted as a standard in the evaluation of the effects of mission variables. Emphasis was given to determination of the influence of entry conditions (velocity and corridor position) on shield weights. Importance was also assigned to the description of the influence of vehicle weight and volume. Examination of the effect of heat-shield material was limited to the extent necessary to show possible benefits accruing from use of advanced materials at the extremes of heat loading. A single configuration was selected to show the influence of allowable heat-shield bond-line temperature. A number of cases were considered in the investigation of the effects of uncertainties in gas

Table I

SCOPE OF HEAT SHIELDING REQUIREMENTS STUDIES

| Parameters / Mission | Vehicle Configuration | Lift-Drag Ratio | Entry Speed (kfs) | Vehicle Volume ($ft^3$) | Vehicle Weight (lb) | Constraints |
|---|---|---|---|---|---|---|
| Earth Lander | Apollo | 0.5 | 36 – 50 | 500 – 1,500 | 5,000 – 15,000 | • Minimum corridor width of 10 sm |
| | M1 | 0.5 | 36 – 65 | 500 – 1,500 | 5,000 – 15,000 | • Maximum deceleration of 10 g |
| | M2 | 1.0 | 36 – 75 | 500 – 1,500 | 5,000 – 15,000 | • Nylon phenolic shield |
| Mars Lander | Apollo | 0.1 | 16 – 40 | 1,000 – 2,500 | 35,000 – 70,000 | • 500° F maximum bond-line temperature |
| | Apollo | 0.5 | 16 – 40 | 1,000 – 2,500 | 35,000 – 70,000 | |
| Mars Orbiter | Apollo | 0.5 | 25 – 35 | 40,000 – 70,000 | 300,000–500,000 | |
| | Blunted Cone | 0.5 | 25 – 35 | 40,000 – 70,000 | 300,000–500,000 | |

emissivity, boundary layer transition, atmospheric composition, and atmospheric structure. A total of approximately 100 distinct situations were examined.

In order to correctly establish the effects of the parameters and uncertainties, point analysis procedures were used. In particular, detailed time histories of the environment and shield response were computed in each case at a number of body stations sufficient to permit accurate evaluation of the total shield weight.

## 5. BASIC DATA GENERATED AND SIGNIFICANT RESULTS

The environmental phenomena experienced by vehicles entering the Martian and terrestrial atmospheres at hyperbolic speeds have been investigated and thermal protection requirements have been examined. Principal results are as follows:

### Flight Mechanics

Entry into the earth atmosphere upon return from a Mars mission can be accomplished by relatively simple maneuvers using the trimmed-lift, roll-control mode. Imposition of a 10g deceleration limit and a 10-sm corridor width requirement restricts the maximum entry velocities with lift/drag ratios of 0.5 and 1.0 to about 57,000 ft/sec and 68,000 ft/sec, respectively.

Atmospheric braking appears to be an efficient means of orbital capture at Mars. With full positive lift to exit the atmosphere after constant altitude deceleration (a non-optimum maneuver), the injection velocity requirements for a 500 km circular orbit is about 300 ft/sec – a small fraction of that required for direct retro from hyperbolic approach velocities.

### Atmosphere Characteristics

The equilibrium composition and thermodynamic properties of air were computed at small intervals over a range of pressures to a temperature of 45,000°R to provide accurate data for heat transfer analyses. An upper-bound emissivity model, accounting for the important ultraviolet-deionization and atomic-line emission processes, was selected from the several disparate results. High temperature thermodynamic and radiative properties were computed for several possible Mars gas mixtures. In the molecular radiation regime, the primary contribution to the radiation is from the CN ultraviolet bands. In the atomic radiation regime, carbon deionization is as important as oxygen and nitrogen deionization. The dependence of the radiance on carbon-dioxide/nitrogen ratio is slight except at velocities below 30,000 ft/sec.

## Heat Transfer Analyses

Several advanced techniques were developed to describe the heating experienced at hyperbolic entry speeds. For the entry situations considered, energy loss from the shock layer by radiative emission significantly alters the flow field. Temperature and velocity near the vehicle surface are reduced and the shock standoff distance is decreased. Most importantly, the radiative heat transfer to the vehicle surface is greatly reduced as shown in Fig. 1 for two representative situations. However, it is noteworthy that the cooling has relatively little effect on the radiative heat transfer

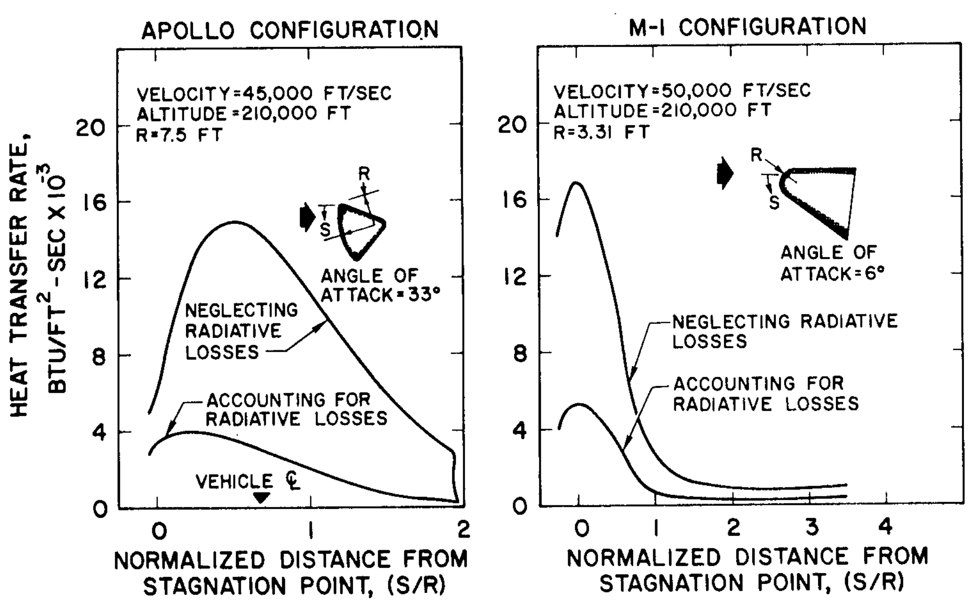


Fig. 1 Influence of Radiative Cooling on Radiative Energy Transfer

distribution normalized with respect to the stagnation value. For the Earth's atmosphere, the distribution remains essentially frozen when velocity and altitude are varied. Conversely, for the Mars atmosphere gas mixture, the radiative heat flux distribution changes appreciably in the velocity range where the transition from the molecular radiation regime to the atomic radiation regime occurs.

The dominant emissions from the Martian and terrestrial gases are from relatively small portions of the spectrum. Thus, self-absorption of radiation in the shock-layer is important in some entry cases considered, even though the optical thickness based on the Planck mean-absorption coefficient is small. Ablation products injected into the boundary layer may further increase the extent of absorption.

Convective heating is appreciably influenced by shock-curvature-induced vorticity. For slender vehicles traveling at high altitudes, the vortical layer (fluid emanating from the near-normal portion of the shock) is entrained by the boundary layer with a resultant increase in convective heating level, as depicted in Fig. 2. On the other hand the injection of gaseous ablation products greatly reduces convective heat load.

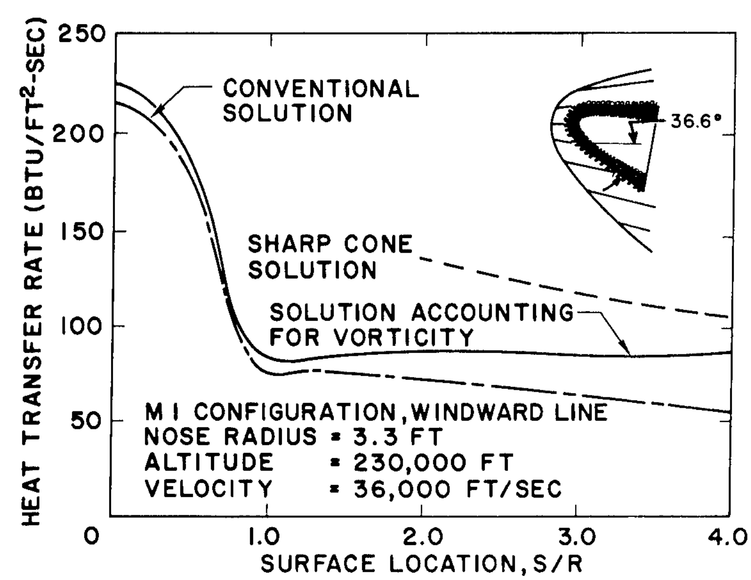


Fig. 2 Influence of Vorticity on Convective Heat Transfer

In fact, injection rates become sufficiently high during periods of intense radiation to cause "blow-off" of the thermal boundary layer and convection ceases.

The radiative cooling and mass transpiration effects preclude the application of simple, closed-form correlation equations for the heat transfer to the vehicle surface. Figure 3 vividly demonstrates that for typical conditions the actual net heat transfer may be an order-of-magnitude less than that computed neglecting energy losses and assuming a cold, non-ablating wall.

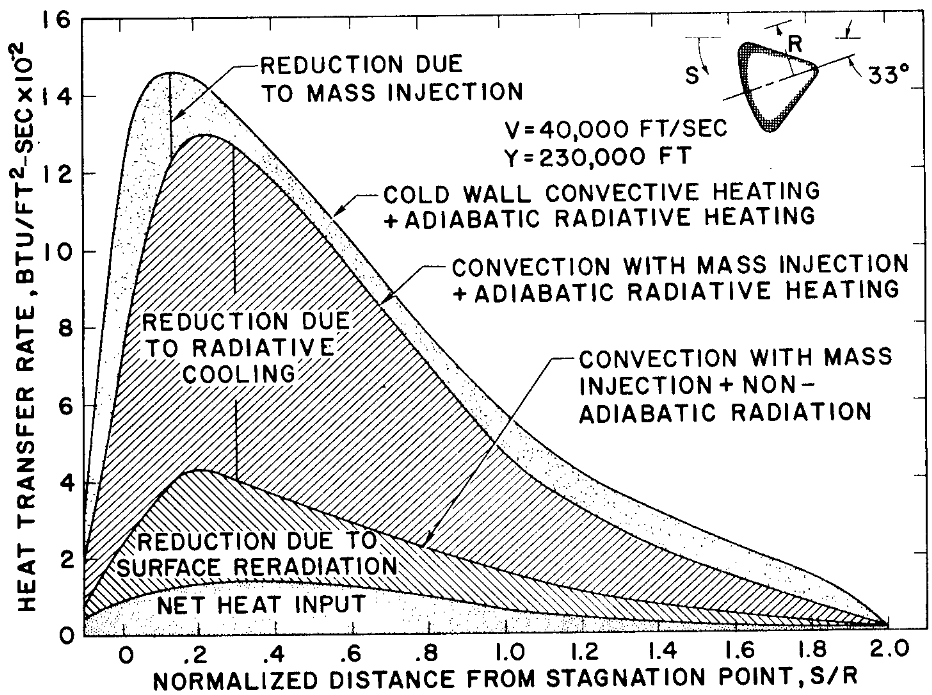


Fig. 3 Comparison of Uncorrected Heat Input With Actual Heat Input

## Material Analyses

An experimentally verified theoretical model for the thermochemical performance in air of nylon-phenolic, the reference material, was available at the outset of this study.

The validity of an extension of the theoretical model for description of material performance in Mars-like gas mixtures has been established in the present work. Thus, the necessity for reliance on gross, empirical techniques for prediction of material responses has been overcome.

The performance theory for charring ablators provides a basis for optimizing the material formulation. In the hyperbolic entry environment a high char density is desirable to minimize erosion. Performance is enhanced by use of high-hydrogen-content organic polymers that evolve high-enthalpy gaseous products upon pyrolysis. A composite of graphite and polyethylene appears attractive from these considerations.

## Vehicle Heat-Shielding Requirements

Atmospheric braking of hyperbolic entry vehicles appears thermally feasible provided that efficient configurations and heat shield materials are selected. Some typical results for the shielding requirements applicable to the Apollo configuration are:

APOLLO SHIELDING REQUIREMENTS

| Mission | Entry Velocity (ft/sec) | Vehicle Volume ($ft^3$) | Vehicle Weight (lb) | Shield Weight (lb) | $\frac{\text{Shield Weight}}{\text{Total Vehicle Weight}}$ (%) |
|---|---|---|---|---|---|
| Earth Landing | 45,000 | 1,000 | 10,000 | 1,500 | 15 |
| Mars Landing | 35,000 | 1,750 | 60,000 | 2,500 | 4 |
| Mars Orbital Capture | 30,000 | 55,000 | 400,000 | 20,000 | 5 |

A comparison of heat-shield weights for various configurations for Earth entry is shown in Fig. 4. It can be seen that the highly blunt Apollo configuration remains attractive to velocities of about 50,000 ft/sec. The use of slender, high lift-to-drag ratio configurations such as the M2 to gain increased corridor width and hence make possible successful entries up to 68,000 ft/sec requires significantly greater heat shield weight.

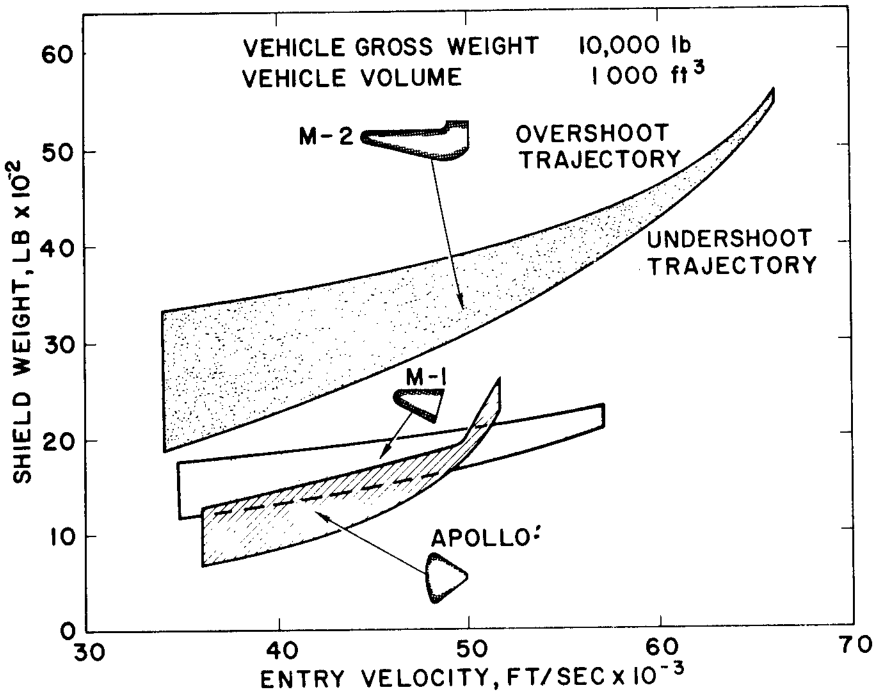


Fig. 4 Comparison of Heat Shielding Requirements for Various Earth Entry Vehicles

For a given configuration, it is found that heat shield weight does not increase drastically with increase of entry velocity over the ranges considered even when radiation dominates. This result is at first surprising, however it is explained in terms of material performance. Figure 5 shows that, for a representative location on the Apollo, a sixfold increase in heat load increases the required shield thickness less than 30 percent. The charring ablator performs more efficiently at higher heat flux levels. With regard to corridor position, shielding requirements are almost always minimized by entering along the undershoot trajectory.

Shield weight for a given payload is minimized by efficient packaging, i.e., reduction of total volume. At a given value of the ballistic coefficient, heat shield section thicknesses do not change appreciably with perturbation of size and volume, and thus the ratio of total heat shield weight to total vehicle surface area may be correlated in terms of the ballistic coefficient alone, as shown in Fig. 6.

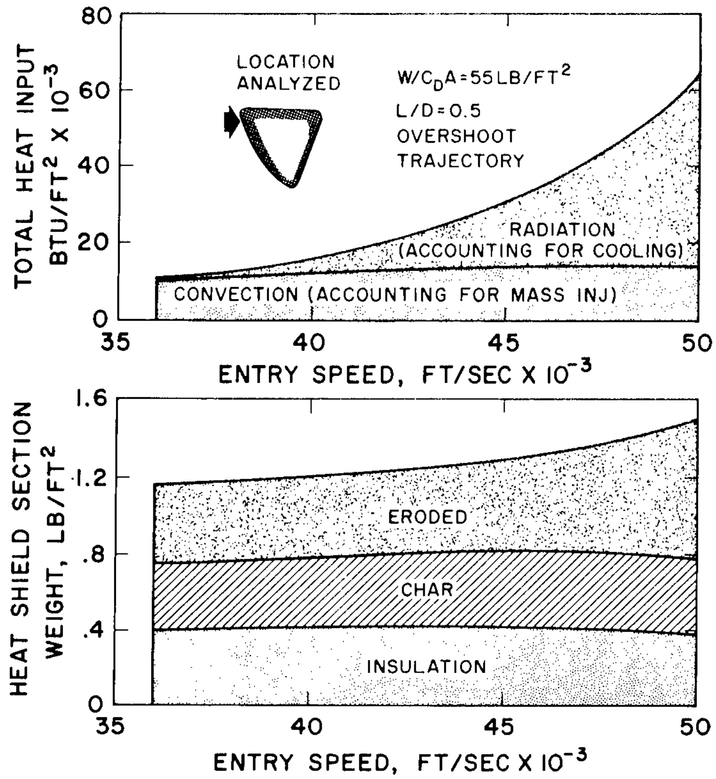


Fig. 5 Relationship Between Heat Input and Heat Shield Requirements with Increasing Entry Speed

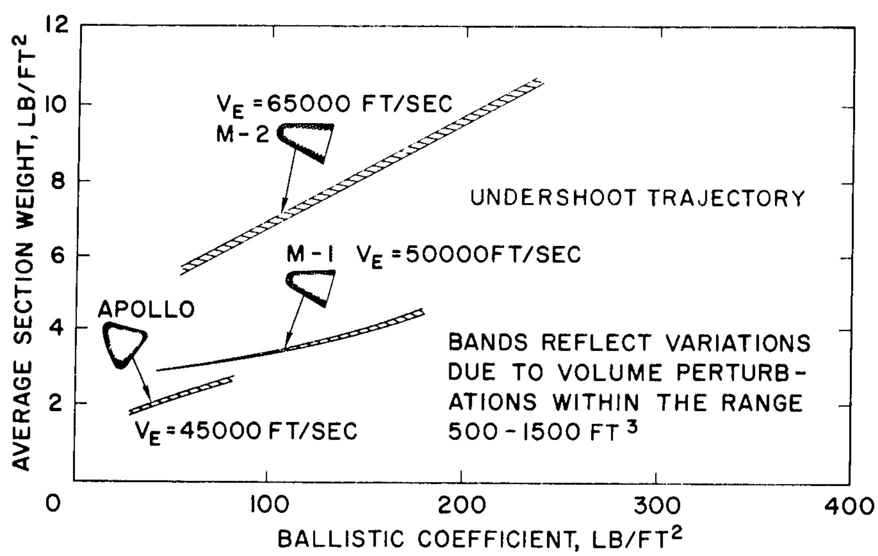


Fig. 6 Influence of Ballistic Coefficient on Heat Shielding Requirements

The effect on shield weight of two of the more prominent environmental uncertainties that have been examined is shown in Fig. 7. Large uncertainties in gas emissivity appear relatively unimportant, at least for the entry situations considered in this study. The uncertainty in the occurrence of boundary layer transition is shown to be of somewhat greater importance. Uncertainties in the structure and composition

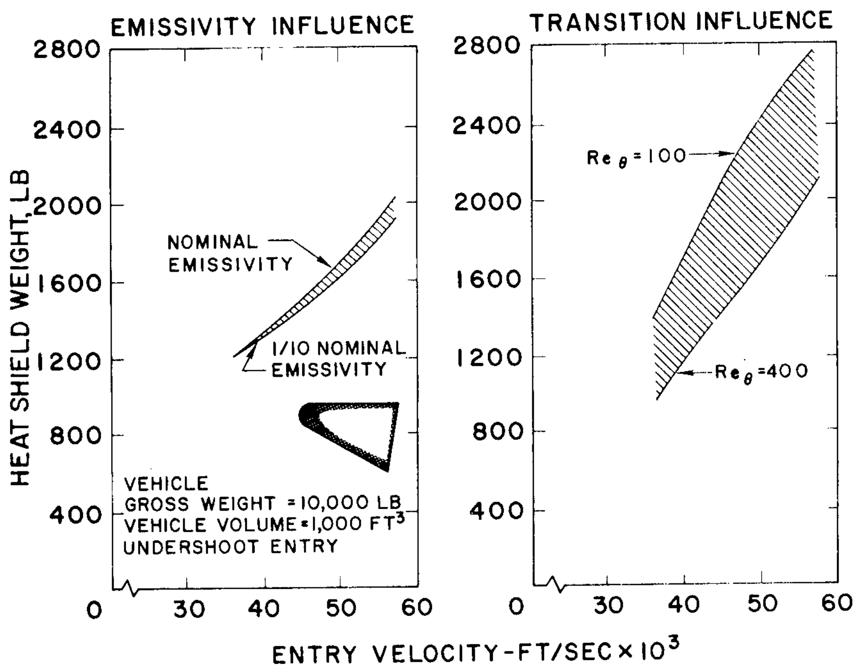


Fig. 7 Influence of Environmental Uncertainties on Heat Shield Requirements for M-1 Vehicle

of the Martian atmosphere are not influential in determining heat shielding requirements for hyperbolic entry. On the other hand, the possible mechanical erosion of heat shielding material would have significant effect. Thick char layers are predicted considering chemical erosion alone. Spallation of char could result in a doubling of heat shield thickness.

## 6. IMPLICATIONS FOR RESEARCH

A number of basic problem areas and significant uncertainties have been encountered in the study. The aerothermal technology would benefit from:

- Continued theoretical and experimental investigations of high temperature gas properties under non-equilibrium and non-ideal conditions
- Formulation of a numerical technique for description of non-adiabatic flow with self-absorption by non-grey gases
- Theoretical evaluation of the effects of ablation products on radiative energy transport both within the boundary layer and in the wake region
- Experimental investigations of the transition phenomena with surface roughness, mass injection, and wall-cooling conditions of flight simulated
- Development of rigorous analytic procedures for the evaluation of vorticity effect on convective heat transfer with mass injection
- Assessment of the effect of precursor radiation on the flow
- Improvement of material formulations guided by theoretical analyses
- Experimental determination of the environmental regimes where mechanical erosion of charring materials will occur to significant degree

## 7. SUGGESTED ADDITIONAL EFFORT

The following recommendations are made for further work in the area of heat-shielding requirements

- Study the effects of geometry variables to establish optimum vehicle configuration
- Investigate the influence of altitude maneuvering and lift modulation to establish optimal entry trajectories
- Conduct more detailed analyses of uncertainties on shielding requirements



NASA-Langley, 1965 CR-308

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